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TECHNICAL MEMORANDUM

X-414

COLD-AIR INVESTIGATION OF A THREE-STAGE TURBINE WITH A BLADE- TO JET-SPEED RATIO OF 0.135 DESIGNED FOR A 20,000-POUND-THRUST HYDROGEN-OXYGEN ROCKET TURBOPUMP APPLICATION

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COLD-AIR INVESTIGATION OF A THREE-STAGE TURBINE WITH A BLADE- TO JET-SPEED RATIO OF 0.135 DESIGNED FOR A 20,000-POUND-THRUST HYDROGEN-OXYGEN ROCKET TURBOPUMP APPLICATION*

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SUMMARY

A three-stage turbine designed to power the propellant pumps of a 20,000-pound-thrust hydrogen-and-oxygen rocket stage was investigated experimentally with cold air. The turbine had an efficiency of 0.54 (based on total- to static-pressure ratio) at design speed and design pressure ratio. The performance results are compared with those obtained in a reference investigation of a one- and two-stage turbine operated at a blade- to jet-speed ratio of 0.135. This comparison showed that the efficiency was increased from 0.31 to 0.54 as a result of increasing the number of stages from one to three. The comparison also showed that the overall efficiency of 0.54 was indicative of the level of performance that might be expected from a three-stage turbine operated at this blade- to jet-speed ratio. Thus, the performance results confirm the analytical trend of the effect of stage number on overall turbine efficiency in this low range of blade- to jet-speed ratio.

INTRODUCTION

The turbine research program at the NASA Lewis Research Center includes turbines for various rocket propellant pump drive and auxiliary power applications. One such application of interest is the propellant pump drive turbine for hydrogen-fueled rockets. A type of turbopump configuration commonly considered for this application is a bleed system, in which a portion of the propellants is bled off at the pump discharge, burned in a gas generator, passed through the turbine, and then exhausted overboard. The design blade- to jet-speed ratios for these turbines are generally lower than those normally encountered because of the high jet velocity of the hydrogen that is in the combustion gas.

*Title, Unclassified.

It is desired to obtain a reasonable turbine efficiency in order to minimize the amount of propellant expended by the turbine. Many analytical studies, such as those of reference 1, have shown that efficiency can be increased by increasing the number of stages in this low blade- to jet-speed ratio range.

In order to experimentally determine the effect of stage number on performance in this low blade- to jet-speed ratio range, a three-stage turbine was investigated. The turbine was designed specifically to meet the requirements of a bleed-system turbopump for a 20,000-pound-thrust hydrogen-oxygen rocket. The turbine was operated with air at inlet conditions of 800° R and 75 pounds per square inch absolute. The pressure ratio was varied over a range for various speeds from 60- to 110-percent design speed.

This report will present the design and cold-air performance evaluation of the three-stage turbine. The effect of stage number will be shown by comparing the performance results with those of reference 2, where the performance is presented for a one- and a two-stage turbine configuration in the same range of blade- to jet-speed ratio.

SYMBOLS

c_p	specific heat at constant pressure, Btu/(lb)(°R)
g	acceleration of gravity, 32.17 ft/sec^2
Δh	specific enthalpy drop, Btu/lb
J	mechanical equivalent of heat, 778.2 ft-lb/Btu
p	static pressure, lb/sq ft
p'	stagnation pressure, lb/sq ft
R	gas constant, (ft-lb)/(lb-°R)
T	stagnation temperature, °R
U	blade velocity, ft/sec
V	absolute gas velocity, ft/sec
V_{cr}	critical velocity, $\left(\frac{2\gamma gRT}{\gamma + 1}\right)^{1/2}$, ft/sec

V_j theoretical jet velocity, $\left\{2gJc_pT_0\left[1 - \left(\frac{p_6}{p_0'}\right)^{\frac{\gamma-1}{\gamma}}\right]\right\}^{1/2}$, ft/sec

W gas velocity relative to rotor, ft/sec

W_{cr} critical velocity relative to rotor blade, ft/sec

w gas-flow rate, lb/sec

γ ratio of specific heats

δ ratio of pressure at turbine inlet to NACA standard pressure, p_0'/p^*

ϵ function of γ, $\frac{\gamma^*}{\gamma}\frac{\left(\frac{\gamma+1}{2}\right)^{\frac{\gamma}{\gamma-1}}}{\left(\frac{\gamma^*+1}{2}\right)^{\frac{\gamma^*}{\gamma^*-1}}}$

η_s turbine efficiency based on total- to static-pressure ratio across turbine

θ_{cr} squared ratio of critical velocity at turbine inlet to critical velocity of NACA standard air, $(V_{cr,0}/V_{cr}^*)^2$

Subscripts:

m mean-radius value

x axial component

0 station at turbine inlet

1 station between first-stage stator and first-stage rotor

2 station between first-stage rotor and second-stage stator

3 station between second-stage stator and second-stage rotor

4 station between second-stage rotor and third-stage stator

5 station between third-stage stator and third-stage rotor

6 station at turbine outlet

Superscript:

$*$ refers to air at NACA standard conditions, $T^* = 518.7^\circ$ R, $p^* = 2116$ lb/sq ft, $\gamma^* = 1.4$

TURBINE DESIGN

As mentioned in the INTRODUCTION the turbine design was based on the requirements of a bleed-system turbopump for a 20,000-pound-thrust hydrogen-oxygen rocket. The requirements of the turbine that evolved from a system study are given in the following table:

	Engine conditions	NACA standard air conditions[a]
Weight flow, w, lb/sec	0.765	0.3079
Specific enthalpy drop, Δh, Btu/lb	1059.6	36.47
Rotative speed, rpm	60,000	11,132
Mean-radius blade speed, U_m, ft/sec	1264.3	234.6
Pressure ratio, p_0'/p_6	9.285	10.332

[a]The corrections from engine conditions to NACA standard air were made according to ref. 3.

This turbine was a three-stage impulse design having a blade- to jet-speed ratio of 0.135 and a mean blading diameter of 4.83 inches. The overall static efficiency estimated for this design was 0.61, and the total efficiencies were 0.71, 0.66, and 0.66 for stages one, two, and three, respectively. These efficiencies were obtained from curves similar to those of reference 1.

The velocity diagrams were laid out to provide an equal work split among the three stages. The state of the gases was computed at the free-stream stations at the turbine inlet and outlet and the four interstage stations. The estimated efficiencies were used to determine the flow area required at the various stations. The resulting velocity diagrams are shown in figure 1. Referring to the figure it is seen that the turning varies from 135° in the first stage to 104° in the third stage. All stages are essentially impulse with high subsonic relative velocities.

It would be expected that the turbine design point is near limiting-loading since the axial critical velocity ratio out of the turbine is 0.65.

The blading that was laid out to operate with this velocity diagram was of nontwisted design, that is, the blade profiles did not change along the radius. A sketch of the rotor-blade passages and profiles is shown in figure 2. The turbine blading geometry is summarized in table I. The blade height varies from 0.212 inch at the inlet to 0.416 inch at the outlet. The minimum thickness trailing edge that could be fabricated was 0.008 inch, which corresponds to a blockage ratio of about 0.20 in the first-stage rotor. Thus, it would be expected that the trailing-edge blockage losses would be higher compared with a larger turbine such as that used to determine the overall loss coefficients used in the efficiency estimation curves.

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As a part of the design procedure an analysis of the flow in the blade channel was made to determine the blade surface velocities using a quasi-three-dimensional method (see ref. 4). The blade surface velocity diagrams are shown in figure 3. The blade loading is quite high for the three rotor-blade passages, especially the second and third stages. The solidity values (solidity is inversely related to blade loading) were 2.52, 2.00, and 2.05 for stages one, two, and three, respectively. For the second and third stages this solidity is lower than that recommended by Zweifel (shown in ref. 5), which was 2.6 for both stages. However, optimum loading was compromised because of fabrication difficulties involved in milling smaller passages to the depth required in the second and third stages.

APPARATUS INSTRUMENTATION AND METHODS

The apparatus and procedure are the same as those described in reference 6 except for the turbine rotor and stator and the accommodating casing pieces. A photograph of the turbine rotor is shown in figure 4, and a diagrammatic sketch of the turbine test section is shown in figure 5. The design rotor tip clearance was 0.011 inch, and the design labyrinth radial clearance was 0.0095 inch at the tongues. The stator blades were shaped in long continuous pieces from bar stock and cut to length afterward. The retention holes in the stator rings were cut by an electric arc destruction method. Each of the three rotor stages was made by milling the passages in a single blank.

The turbine was instrumented with eight static taps each at the inlet and outlet (stations 0 and 6, fig. 5) which were spaced 90° apart on the inner and outer walls. The inlet gas temperature was measured slightly upstream of station 0 with three thermocouples. The turbine shaft torque was measured with a commercial self-balancing torque cell, and the shaft speed was measured with an electronic events-per-unit-time meter. The turbine airflow was metered with an ASME flat plate orifice.

The turbine was operated with inlet conditions maintained at 800° R and 75 pounds per square inch absolute. The ratio of inlet total pressure to outlet static pressure was used in rating the turbine. The actual turbine specific work was computed from the torque, speed, and weight-flow measurements. The data were obtained over a range of speed from 60 to 110 percent of design speed. At each speed the pressure ratio was varied over a range from approximately 5 to 20. E-635

RESULTS AND DISCUSSION

The overall performance data (corrected to inlet conditions of NACA standard air) obtained with the turbine are shown in figure 6. Figure 6(a) shows the variation of corrected weight flow over the range of speed and pressure ratio. It can be seen that the turbine is choked at all speeds since weight flow does not vary with pressure ratio. It can also be concluded that a rotor is choked since corrected weight flow varies with rotor speed. The corrected weight flow at design speed and pressure ratio was 0.318 pound per second, which was 1.03 design weight flow. This deviation could occur as a result of tolerances in the blade fabrication. In the first-stage rotor, for example, a deviation of 0.001 inch in throat area dimension results in a 3-percent error in design throat area. However, it would be expected that performance characteristics would be virtually unaffected by this small change in mass flow.

The variation of corrected enthalpy drop is shown in figure 6(b). The work output at design speed and pressure ratio was 32.3 Btu per pound, which was 0.885 of the design value of 36.47 Btu per pound. The trend of the curve at design speed confirms the proximity to limiting-loading mentioned in the TURBINE DESIGN section. The limiting-loading work output was 1.02 of that at design pressure ratio.

The efficiencies obtained with the turbine over the range of speed and pressure ratio are shown in figure 6(c). The maximum efficiency obtained was 0.60 at 1.10 design speed and at a pressure ratio of 6. The efficiency at design speed and pressure ratio was 0.54.

The performance results are shown in figure 7 in terms of blade- to jet-speed ratio. Dashed lines are shown through design pressure ratio and a pressure ratio of 17. The constant-pressure-ratio lines depict the usual trend of efficiency as a function of blade- to jet-speed ratio. Usually a single line is found to correlate turbine performance data on this basis with a certain amount of scatter. For the subject turbine the individual speed lines deviate from any given pressure-ratio line toward a more vertical orientation. This is because the turbine is at or near limiting-loading for pressure ratios of 10 or above (see fig. 6(b)). The design-pressure-ratio line crosses the design-speed line at a blade- to jet-speed ratio of 0.135 and at an efficiency value of 0.54.

The effect of number of stages on overall turbine efficiency is shown in figure 8 for the design value of blade- to jet-speed ratio (0.135). The one- and two-stage configurations were obtained from reference 2. The efficiency increases from 0.31 for the one-stage configuration to 0.54 for the subject three-stage turbine. This trend agrees with that which could be determined from analytical studies such as reference 1. It would be expected that the increase in efficiency with increasing number of turbine stages would diminish as more stages were added. However, it is indicated that utilizing a four-stage turbine design would result in a substantial increase in overall efficiency. For the specific application the weight of the turbopump system as well as the efficiency must be considered to determine the ultimate merit of a given configuration. However, the experimental results as well as the analytical trends indicate the importance of multistaging in order to achieve reasonable efficiency in this low range of blade- to jet-speed ratio.

As mentioned previously, the efficiency at design conditions was 0.54, which was 7 points lower than that used in the design procedure. Related to the rocket application, these performance results would require that the bleed rate be increased from 0.0135 to 0.0152. The difference between the experimental efficiency and that used in the design procedure can be attributed in part to the approximate nature of the curves used to select the design efficiency as well as certain geometric factors such as relatively large trailing-edge blockage, large ratio of tip clearance to blade height, and leakage through the interstage labyrinth seals that bypasses the stators. In addition, the high blade loading used in the second- and third-stage rotors may cause some impairment to performance level. However, it can be said, based on the comparison with the results of reference 2, that the efficiency of the turbine represents the level of performance that might be expected from a turbine of this size operating at a blade- to jet-speed ratio of 0.135.

SUMMARY OF RESULTS

A three-stage turbine designed to drive the propellant pumps of a 20,000-pound-thrust hydrogen-oxygen rocket was investigated in cold air. The objective was to determine the effect of multistaging for a low blade- to jet-speed ratio of 0.135. The results are as follows:

1. The turbine efficiency at design speed and pressure ratio was 0.54. This efficiency was felt to be indicative of the performance level that might be expected for a three-stage turbine operating at a blade- to jet-speed ratio of 0.135.

2. A comparison of the performance results with those obtained from a reference one-stage and a two-stage turbine investigation showed that considerable improvement in performance was obtained by increasing the number of stages, thus confirming the analytically predicted trend.

Lewis Research Center
National Aeronautics and Space Administration
Cleveland, Ohio, August 22, 1960

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2. Kofskey, Milton G.: Performance Evaluation of a Two-Stage Turbine Designed for a Ratio of Blade Speed to Jet Speed of 0.146. NASA TM X-146, 1959.

3. Heaton, Thomas R., Slivka, William S., and Westra, Leonard F.: Cold-Air Investigation of a Turbine with Nontwisted Rotor Blades Suitable for Air Cooling. NACA RM E52A25, 1952.

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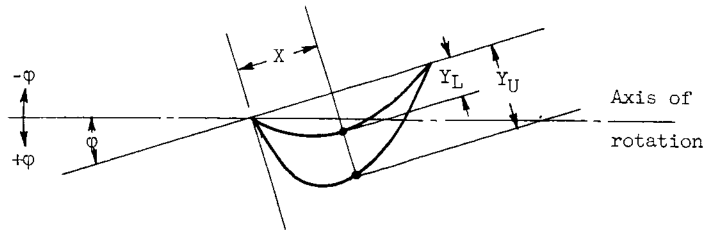
5. Miser, James W., Stewart, Warner L., and Whitney, Warren J.: Analysis of Turbomachine Viscous Losses Affected by Change in Blade Geometry. NACA RM E56F21, 1956.

6. Whitney, Warren J., and Wintucky, William T.: Experimental Investigation of a 7-Inch-Tip-Diameter Transonic Turbine. NACA RM E57J29, 1958.

TABLE I. - BLADE COORDINATES

(a) Stator

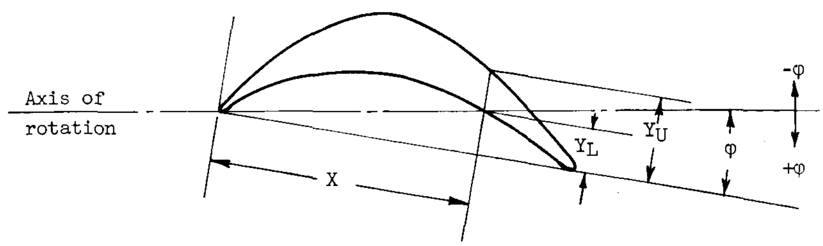


First stage			Second stage			Third stage		
Orientation angle, φ								
53°5'			17°45'			13°41'		
Average blade length								
0.212			0.255			0.327		
Number of blades								
30			52			56		
X	Y_U	Y_L	X	Y_U	Y_L	X	Y_U	Y_L
0.000	0.010	0.010	0.000	0.004	0.004	0.000	0.004	0.004
.010	.034	.000	.004	-----	.000	.020	-----	.023
.050	.093	.027	.020	-----	.038	.040	-----	.056
.100	.137	.056	.040	-----	.084	.060	-----	.084
.150	.164	.078	.060	-----	.119	.080	-----	.105
.200	.178	.093	.075	[a].229	-----	.098	[a].173	-----
.250	.182	.103	.080	.246	.146	.100	-----	.122
.300	.176	.108	.100	.287	.166	.120	.206	.135
.350	.165	.109	.120	.313	.181	.140	.228	.146
.400	[a].151	.106	.140	.329	.194	.160	.241	.153
.450	-----	.100	.160	.338	.203	.180	.251	.153
.500	-----	.092	.180	.342	.210	.200	.255	.162
.550	-----	.083	.200	.341	.214	.220	.256	.163
.600	-----	.072	.220	.335	.216	.240	.252	.162
.650	-----	.060	.240	.325	.216	.260	.245	.159
.700	-----	.047	.260	.312	.213	.280	.235	.154
.750	-----	.032	.280	.294	.208	.300	.221	.149
.800	-----	.017	.300	.273	.201	.320	.204	.142
.858	-----	.000	.320	[a].248	.192	.340	.184	.133
.863	.005	.005	.340	-----	.180	.350	[a].174	-----
			.360	-----	.166	.360	-----	.123
			.380	-----	.151	.380	-----	.111
			.400	-----	.132	.400	-----	.098
			.420	-----	.112	.420	-----	.083
			.440	-----	.090	.440	-----	.067
			.460	-----	.067	.460	-----	.049
			.480	-----	.044	.480	-----	.030
			.500	-----	.020	.500	-----	.010
			.524	.005	.005	.516	.005	.005

[a]Straight line from this point to point of tangency with leading- or trailing-edge circle.

TABLE I. - Concluded. BLADE COORDINATES

(b) Rotor



First stage			Second stage			Third stage		
Orientation angle, φ								
$0^\circ 38'$			$-0^\circ 27'$			$1^\circ 52'$		
Average blade length								
0.218			0.288			0.390		
Number of blades								
153			81			69		
X	Y_U	Y_L	X	Y_U	Y_L	X	Y_U	Y_L
0.000	0.004	0.004	0.000	0.004	0.004	0.000	0.004	0.004
.004	-----	.000	.020	-----	.024	.020	-----	.020
.010	-----	.007	.040	-----	.054	.040	-----	.046
.020	-----	.027	.060	-----	.079	.060	-----	.069
.030	-----	.043	.080	[a].145	-----	.080	-----	.087
.033	[a].070	-----	.080	-----	.098	.097	[a].135	-----
.040	.084	.057	.100	.173	.113	.100	-----	.103
.050	.101	.068	.120	.192	.124	.120	.161	.115
.060	.115	.177	.140	.204	.133	.140	.178	.126
.070	.125	.084	.160	.211	.137	.160	.190	.134
.080	.133	.090	.180	.214	.140	.180	.198	.139
.090	.139	.094	.200	.212	.139	.200	.201	.143
.100	.143	.097	.220	.205	.136	.220	.200	.144
.110	.146	.099	.240	.193	.130	.240	.195	.142
.120	.147	.100	.260	.175	.120	.260	.187	.139
.130	.146	.099	.280	.150	.108	.280	.174	.134
.140	.143	.098	.290	[a].136	-----	.300	.157	.125
.150	.140	.095	.300	-----	.092	.320	.138	.114
.160	.135	.092	.320	-----	.070	.340	-----	.100
.170	.128	.087	.340	-----	.044	.342	[a].116	-----
.180	.119	.081	.360	-----	.014	.360	-----	.084
.190	.107	.074	.375	.004	.004	.380	-----	.065
.200	.093	.065				.400	-----	.045
.206	[a].082	-----				.420	-----	.025
.210	-----	.054				.440	-----	.003
.220	-----	.040				.450	.004	.004
.240	-----	.007						
.246	-----	.000						
.250	.004	.004						

[a]Straight line from this point to point of tangency with leading- or trailing-edge circle.

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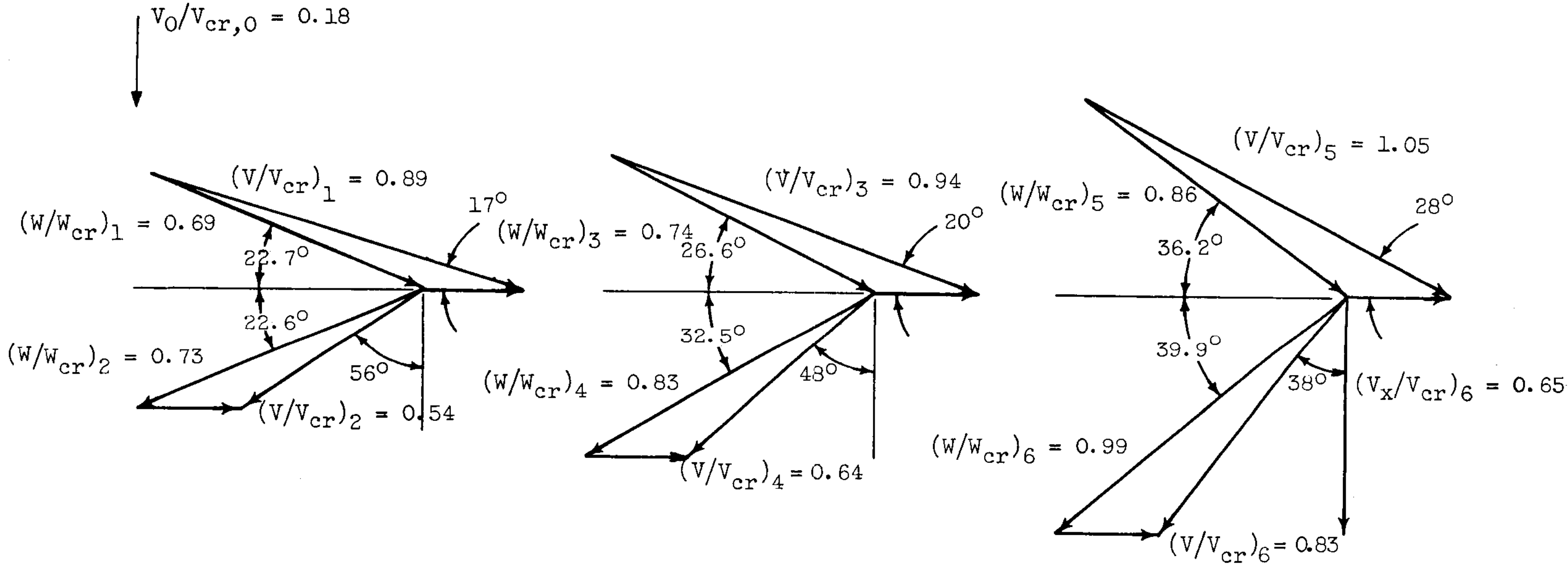

Figure 1. - Velocity diagrams of three-stage turbine for driving propellant pumps of 20,000-pound-thrust hydrogen-oxygen rocket.

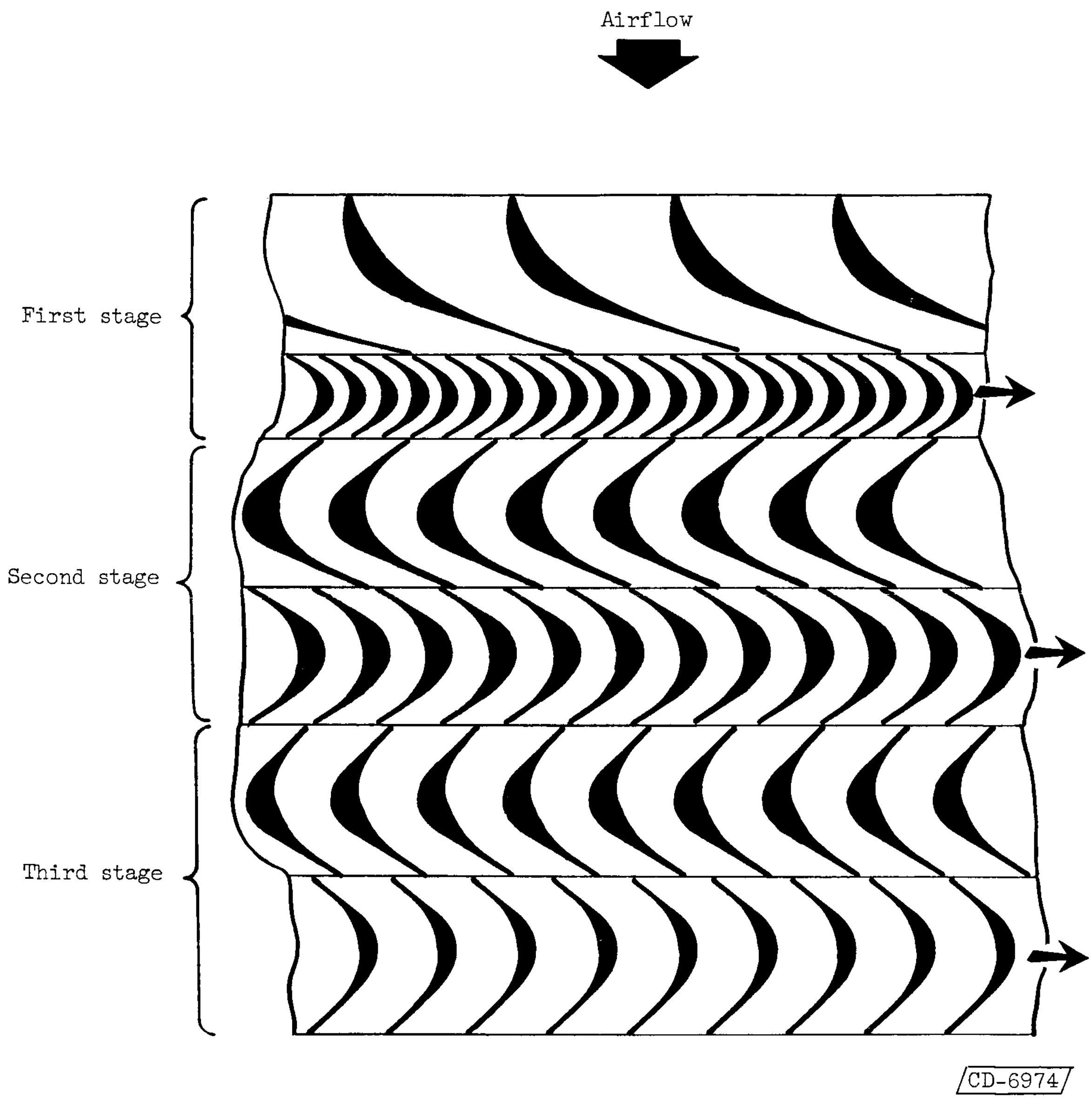

Figure 2. - Turbine rotor and stator blading passages and profiles.

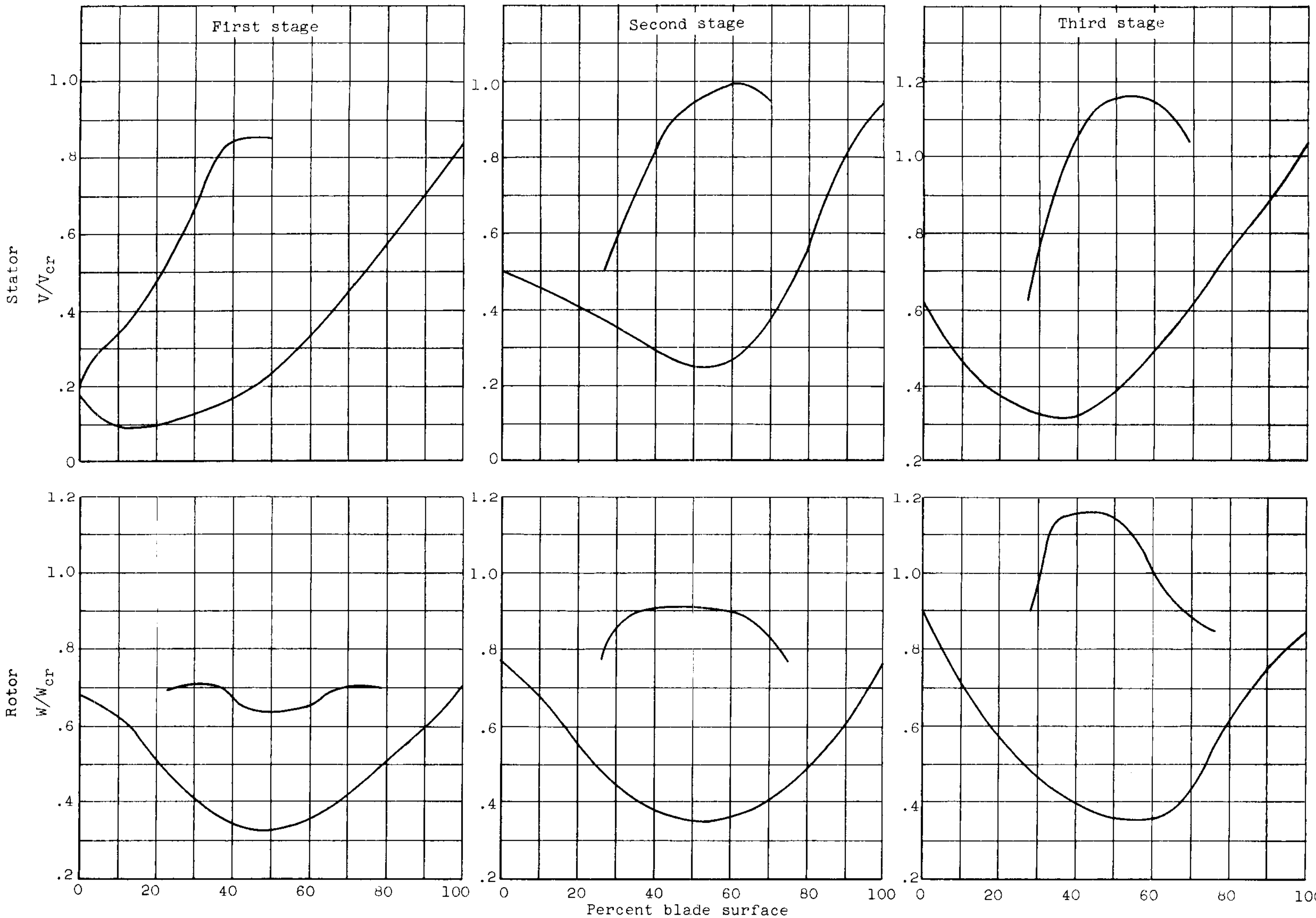

Figure 3. - Blade surface velocity distribution for 20,000-pound-thrust hydrogen-oxygen rocket pump drive turbine.

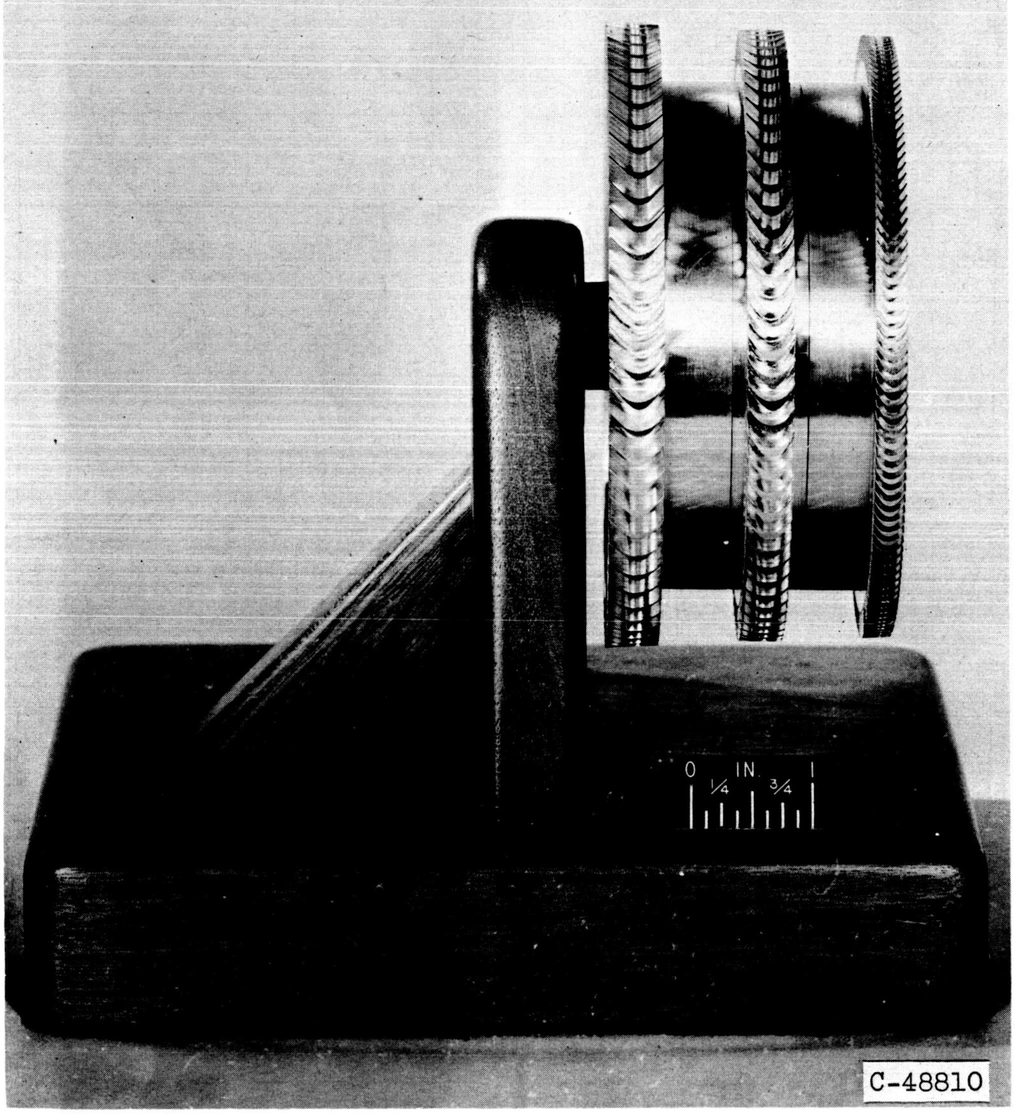

Figure 4. - Turbine rotors.

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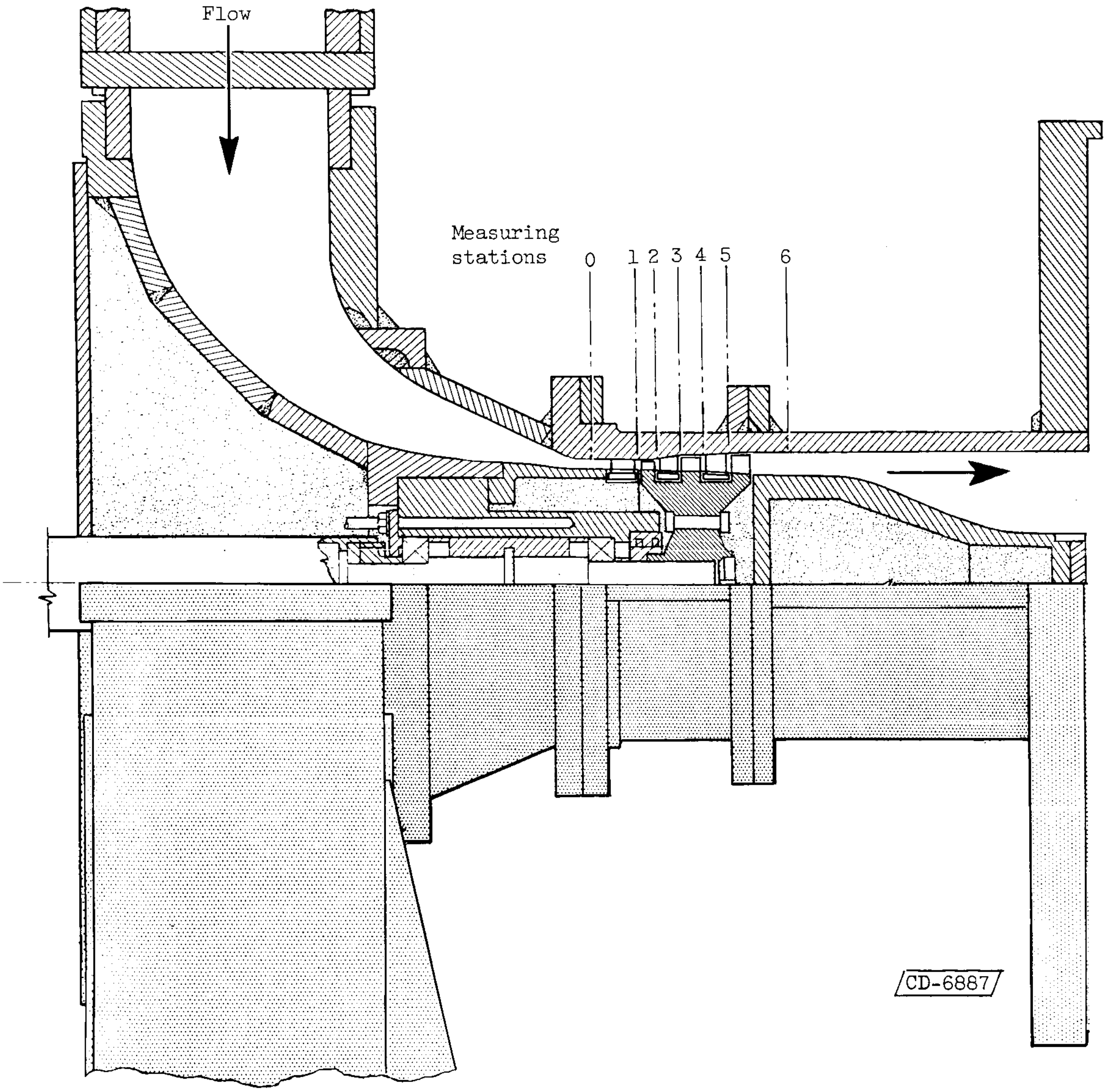
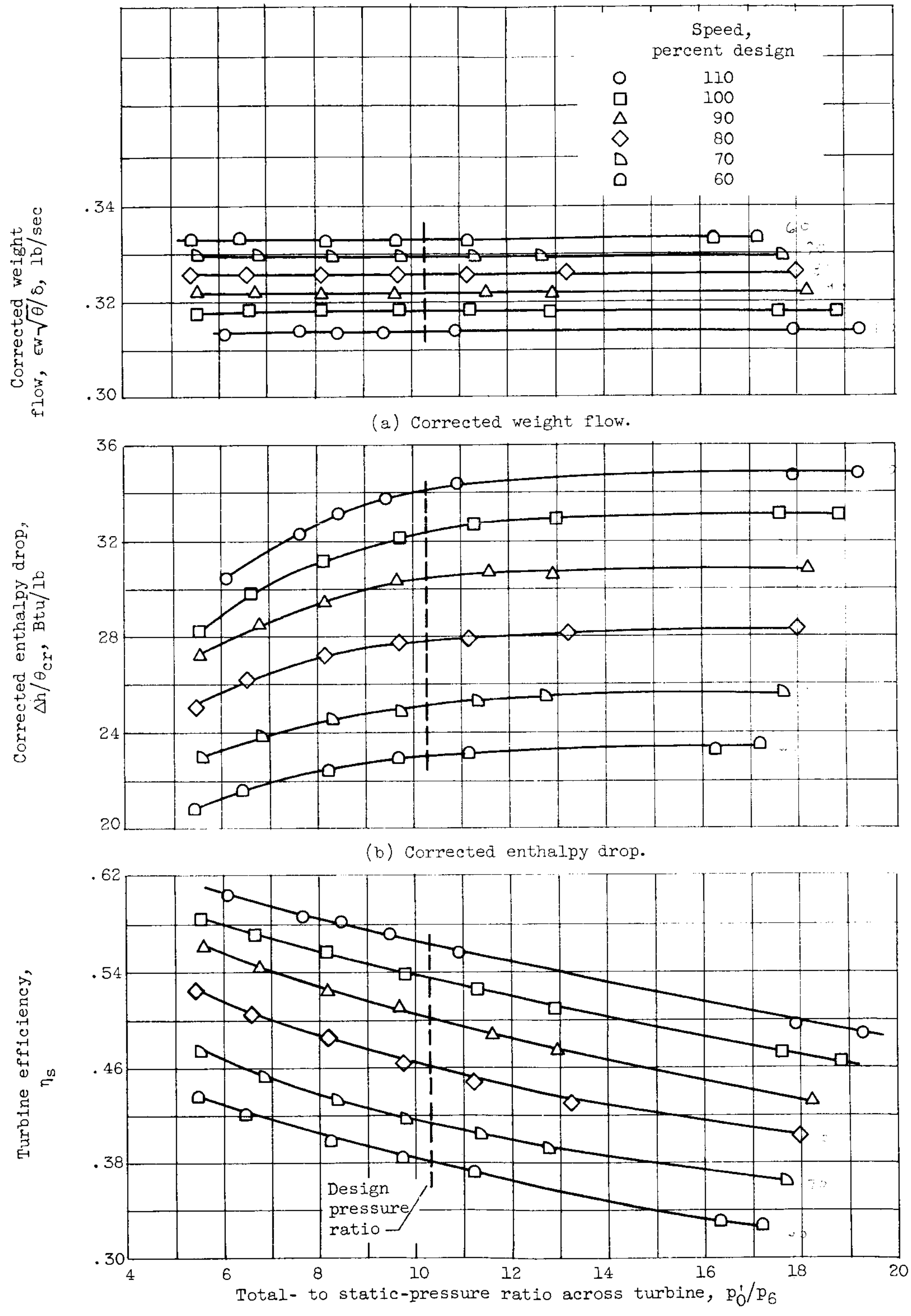


Figure 5. - Diagrammatic sketch of turbine test section.



(a) Corrected weight flow.

(b) Corrected enthalpy drop.

(c) Turbine efficiency.

Figure 6. - Overall turbine performance corrected to inlet conditions of NACA standard air.

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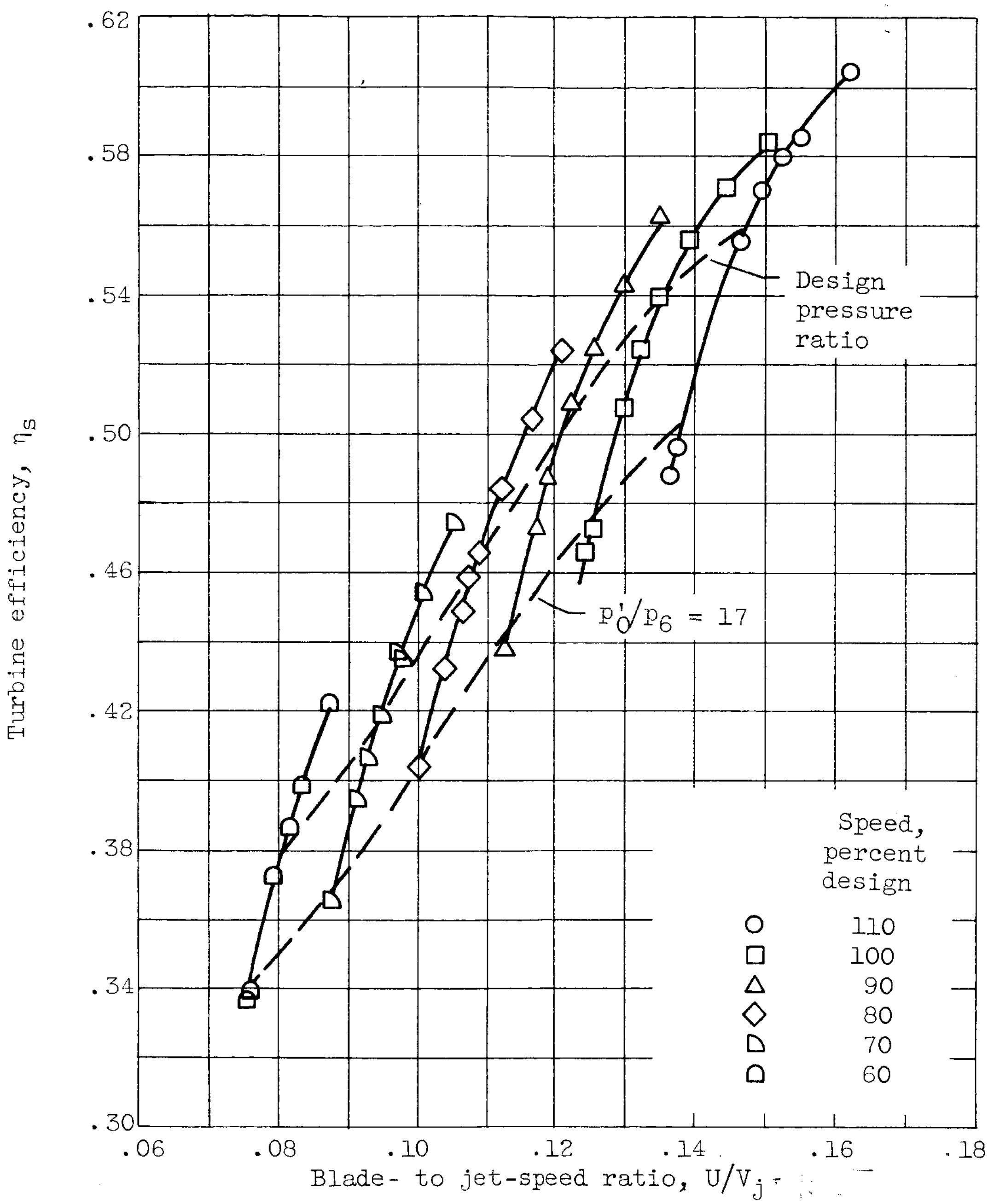

Figure 7. - Overall turbine performance in terms of efficiency blade- to jet-speed ratio characteristics.

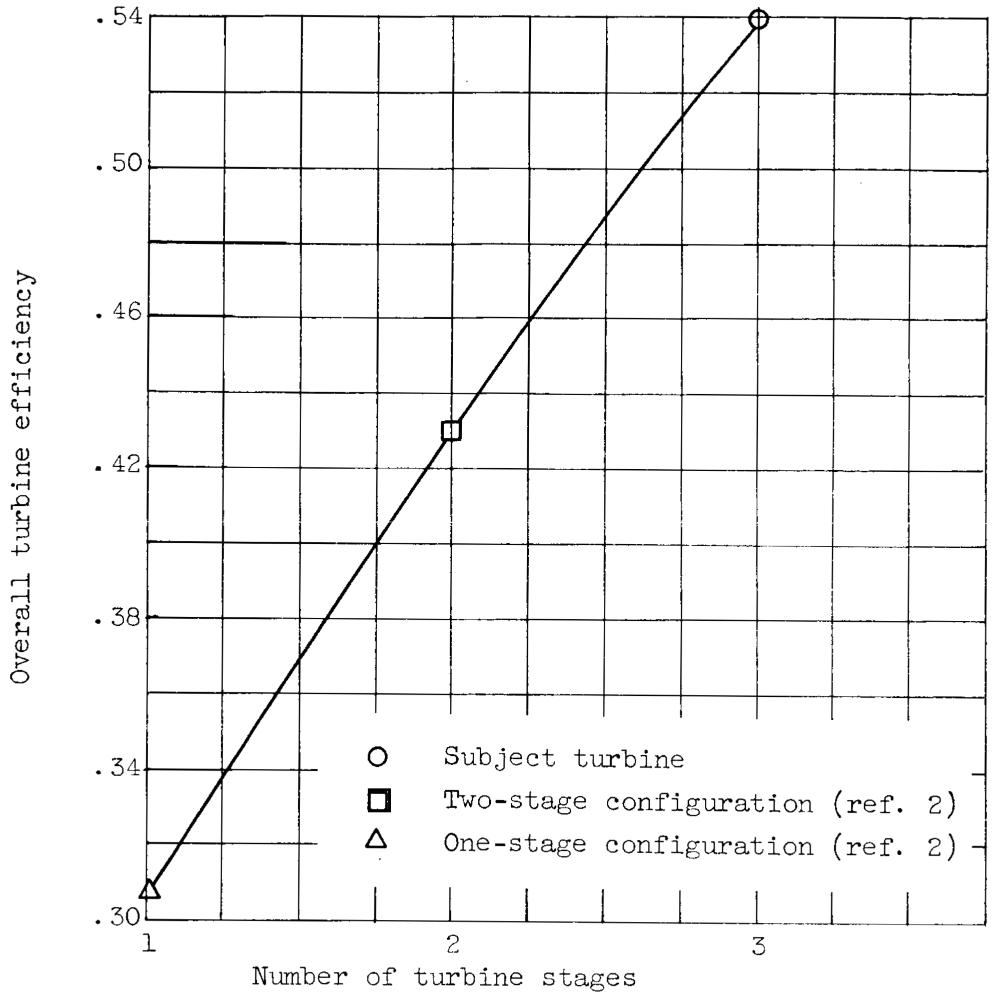

Figure 8. - Effect of number of stages on overall turbine efficiency for a blade- to jet-speed ratio of 0.135.

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